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RESEARCH MEMORANDUM

EFFECTS OF SPANWISE THICKNESS VARIATION ON THE TRANSONIC AERODYNAMIC CHARACTERISTICS OF WINGS HAVING 35° OF SWEEPBACK, ASPECT RATIO 4, AND TAPER RATIO 0.60

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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EFFECTS OF SPANWISE THICKNESS VARIATION ON THE TRANSONIC AERODYNAMIC CHARACTERISTICS OF WINGS HAVING 35° OF SWEEPBACK, ASPECT RATIO 4, AND TAPER RATIO 0.60

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SUMMARY

An investigation has been conducted in the Langley high-speed 7- by 10-foot tunnel to determine the effects of a spanwise variation in thickness ratio on the aerodynamic characteristics of a wing having a plan form identical to that of a constant-percent thickness-ratio wing previously investigated as part of an extensive transonic research program. The wing with thickness variation had 35° of sweepback, aspect ratio 4, and taper ratio 0.60, with airfoil section (parallel to the free stream) tapered from an NACA 65A006 section at the root chord to an NACA 65A002 section at the tip chord. The test Mach number range was from 0.60 to 1.08 at Reynolds numbers of the order of 650,000.

The results of this investigation indicate no important differences in minimum-drag characteristics of the tapered-in-thickness-ratio and constant 6-percent-thick wings, except for a less rapid drag rise through the transonic speed range for the wing tapered in thickness ratio. Drag due to lift for the wing tapered in thickness ratio is higher at moderate lift coefficients at Mach numbers of 0.80 and 1.00. The tapered 6- to 2-percent-thick wing exhibited an appreciably higher lift-curve slope above $M = 0.85$ and through the remaining test Mach number range than the constant 6-percent-thick wing. Subsonic theoretical values of lift-curve slope, aerodynamic center, and lateral center of lift are in fairly good agreement with experiment. Agreement between experiment and theory for these parameters at a Mach number of about 1.00 and higher is generally poor.

INTRODUCTION

As part of an extensive transonic research program, a limited investigation has been conducted in the Langley high-speed 7- by 10-foot tunnel to determine the aerodynamic characteristics of tapered-in-thickness-ratio wings identical in plan form to some of the constant-thickness-ratio wings investigated under this transonic program.

The wing of this investigation had 35° of sweepback, aspect ratio 4, taper ratio 0.60, and an NACA 65A006 airfoil section at the root chord tapered in thickness by straight-line elements to an NACA 65A002 airfoil section at the tip chord. This wing was investigated as a reflection-plane model over a Mach number range from 0.60 to 1.08 through an angle-of-attack range from -6° to 14°. Results of a previous investigation of the effects of thickness taper on 35° and 45° sweptback wings of aspect ratio 6 are given in reference 1.

This paper presents the experimental results of this investigation and gives a brief analysis of the data in conjunction with data obtained from a previous investigation (reference 2) of a wing with the same plan form but with a constant section thickness ratio of 6 percent. Theoretical comparisons are made with experimental values at subsonic and low-supersonic speeds of lift-curve slope, aerodynamic center, and lateral center of lift.

COEFFICIENTS AND SYMBOLS

All force and moment data presented are referred to the wind axes.

C_L lift coefficient (Twice semispan lift/qS)

C_D drag coefficient (Twice semispan drag/qS)

C_m pitching-moment coefficient referred to $0.25\bar{c}$ (Twice semispan pitching moment/$qS\bar{c}$)

C_B bending-moment coefficient due to lift about root chord $\left(\text{Root bending moment}/q\,\frac{S}{2}\,\frac{b}{2}\right)$

$C_{D_{min}}$ minimum drag coefficient $\left(\text{drag coefficient at } C_L = 0\right)$

ΔC_D drag coefficient due to lift $\left(C_D - C_{D_{min}}\right)$

q effective dynamic pressure over span of model, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

V	free-stream velocity, feet per second
ρ	mass density of air, slugs per cubic foot
S	twice wing area of semispan model, square feet
$\bar{c}$	mean aerodynamic chord of wing using theoretical tip, feet $\left(\frac{2}{S}\int_0^{b/2} c^2\ dy\right)$
c	local wing chord, feet
b	twice span of semispan model, feet
t	maximum local section thickness, feet
t/c	airfoil-section thickness ratio
E	modulus of elasticity in bending, pounds per square inch
y	spanwise distance from plane of symmetry, feet
M	effective Mach number over span of model
M_l	local Mach number
M_a	average chordwise local Mach number
y_{cp}	lateral center of lift, percent semispan $\left(100\ \frac{\partial C_B}{\partial C_L}\right)$
α	angle of attack, degrees
α_D	local angle of streamwise twist (negative values indicate decreased α), degrees
$\frac{\alpha_D}{qC_L}$	local twist parameter

MODELS AND METHODS

The steel wing semispan model had 35° of sweepback referred to the quarter-chord line, aspect ratio 4, taper ratio 0.60, and an NACA 65A006 airfoil section at the root chord measured parallel to the free stream joined by straight-line elements to an NACA 65A002 airfoil section at the tip chord. A plan-form drawing of the model is presented in figure 1 and the variation of thickness ratio along the model semispan is presented in figure 2.

This investigation was conducted in the Langley high-speed 7- by 10-foot tunnel. As a means of testing the semispan model at subsonic and low-supersonic Mach numbers in a region outside the tunnel boundary layer, a plate was mounted about 3 inches from the tunnel wall, as shown in figure 3, to produce an effective reflection plane. The reflection-plane boundary layer was such that a velocity equal to 95 percent of the free-stream velocity in the testing region was reached at a distance of 0.16 inch from the surface at the balance center line for all test Mach numbers. This distance represents about 3.8 percent of the model semispan.

At Mach numbers below 0.95 there was practically no velocity gradient in the vicinity of the model. At higher test Mach numbers, however, both chordwise and spanwise Mach number gradients were evident. The variations of local Mach number in the vicinity of the model location are shown in figure 4. The effective Mach numbers were obtained using the relationship

$$M = \frac{2}{S} \int_0^{b/2} cM_a \, dy$$

For the subject wing a spanwise Mach number gradient of generally less than 0.03 was obtained up to and through a Mach number of 1.08. The chordwise gradient reached a maximum value of less than 0.04 at the highest test Mach number.

Spanwise Mach number gradients over the semispan of the comparison wing, which was investigated on the original transonic bump, ranged from 0.06 at subsonic Mach numbers to 0.08 at the highest test Mach number. The maximum chordwise gradient was 0.01. It has been found that no large or consistent differences are shown from the test results of identical wings tested on the original transonic bump and on the reflection-plane setup (reference 3). A discussion of many of the factors that must be considered in the evaluation of bump and reflection-plane tests can be found in reference 3.

Forces and moments were measured by means of an electrical-strain-gage balance system which was mounted outside the tunnel test section. Leakage through a small clearance gap between the turntable (located flush with the reflection-plane surface) and the wing root was restricted by a sponge seal attached to the wing butt and lightly touching the inside of the turntable. It is difficult to isolate the effects of this sponge seal on the forces measured; however, since the only movements of the wing relative to the reflection plane which would produce restraining forces in the sponge seal are the extremely small deflections in the strain gages and balance linkages, it is believed that the restraint to shear within the sponge seal would be a very small portion of the forces being measured. Angle of attack was measured by means of a slide-wire potentiometer. The variation of mean Reynolds number, based on $\bar{c}$, is shown in figure 5 for both the subject and comparison wings.

In view of the small size of the model relative to the effective flow field, jet-boundary and blockage corrections were believed to be insignificant and hence were not applied.

In order to determine the aeroelastic qualities of the wings used in the analysis of this paper, static loads were applied to the wings at two spanwise locations on the quarter-chord lines and the variation of the angle of streamwise twist was measured at four spanwise locations. These loads were applied at the loading points indicated on figure 6 and in proportions which were intended to simulate roughly the theoretical span loading.

THEORETICAL CONSIDERATIONS

Theoretical calculations of lift-curve slope, aerodynamic center, and lateral center-of-lift locations for subsonic and low-supersonic speeds were made, by the same methods used in reference 1, to provide comparisons with the test results. The theoretical parameters were corrected to the elastic condition by the strip-theory method used in reference 1. Since the difference in the aeroelastic corrections for the subject wing and the comparison wing was negligible, only one theoretical curve is shown for the various aerodynamic parameters presented for the two wings.

RESULTS AND DISCUSSION

Presentation of Results

The basic data of the present investigation are presented in figure 7. Summary plots, including comparisons of aerodynamic characteristics with those of the constant 6-percent-thick wing of reference 2, are presented in figures 8 and 9. Slopes presented in the summary figures were measured through zero lift up to a lift coefficient where obvious departure from linearity occurred.

Lift Characteristics

The variation at low lift coefficients of lift-curve slope with Mach number is presented in figure 9 for both the tapered 6- to 2-percent-thick and constant 6-percent-thick wings. The most noticable effects of this spanwise thickness variation on the low-lift characteristics of the plan form investigated are the appreciably greater lift-curve slope of the tapered-in-thickness-ratio wing over the constant 6-percent-thick wing in the high-subsonic and low-supersonic speed range and the higher Mach number at which the maximum lift-curve slope of the tapered configuration occurred. The higher lift-curve slope for the tapered 6- to 2-percent-thick wing seems to be a direct result of aerodynamic effects, since the aeroelastic effects on the lift slopes of both wings are of a relatively small magnitude. Agreement between experiment and theory is fairly good at subsonic speeds but generally poor at the higher test Mach numbers. The lateral center-of-lift locations (fig. 9) for the 6- to 2-percent-thick and constant 6-percent-thick wings are essentially the same throughout the test Mach number range and show very good agreement with theoretical values up to a Mach number of approximately 0.90.

Drag Characteristics

Minimum-drag characteristics for the subject and comparison wing are presented in figure 9. These characteristics are essentially the same except for a less rapid rise of $C_{D_{min}}$ through the transonic speed range for the tapered-in-thickness-ratio wing. Some indication of the accuracy of the minimum drag can be gained from reference 3, figure 21, in which reflection-plane and rocket-model data are compared for two wings. The rocket-model data presented represent wing-plus-interference drag, but, since the fuselage for the configuration was a cylindrical body, it is believed the wing-plus-interference drag is a valid indication of wing-alone drag. Agreement between the minimum-drag results for the rocket models and reflection-plane models is very good. The predicted value of the pressure drag for the tapered-in-thickness-ratio wing at $M = 1.08$ is presented in figure 9. It can be seen that this predicted value is somewhat lower than that obtained from experiment. Drag due to lift, ΔC_D, as presented in figure 8(b), shows that, at lift coefficients ranging from about 0.2 to 0.6 for Mach numbers of 0.80 and 1.00, values of drag due to lift are somewhat higher for the tapered-in-thickness wing. This increase in drag coefficient due to lift for the tapered 6- to 2-percent-thick wing over the constant 6-percent-thick wing may be a result of a more pronounced leading-edge separation and resulting loss of leading-edge suction common to thin airfoils.

Pitching-Moment Characteristics

Subsonic theoretical and experimental values of aerodynamic-center location (fig. 9) referred to $\bar{c}/4$ (positive values of $\partial C_m/\partial C_L$ forward of $\bar{c}/4$) are in fairly good agreement for the tapered 6- to 2-percent-thick wing. From this figure it can be seen that there is a fairly smooth variation in aerodynamic-center location through the transonic speed range for either the constant 6-percent-thick or tapered 6- to 2-percent-thick wing. Pitching-moment characteristics for both wings at high lifts and a Mach number of 1.00 (fig. 8(c)) show no unstable trends up to the highest lift coefficients investigated and a very linear variation with lift coefficient.

CONCLUSIONS

Wind-tunnel tests have been made at low Reynolds numbers to determine the aerodynamic characteristics at transonic speeds of an aspect-ratio-4, 35° sweptback wing having its thickness ratio tapered from 6 percent at the root to 2 percent at the tip. These data are compared with results obtained for a wing of identical plan form but of a constant 6-percent-thickness ratio. The following conclusions were drawn from these comparisons:

1. The minimum-drag characteristics for the tapered 6- to 2-percent-thick and constant 6-percent-thick wings are essentially the same except for a less rapid rise of minimum drag coefficient through the transonic speed range for the tapered-in-thickness-ratio wing.

2. At moderate lift coefficients ranging from 0.2 to 0.6 for Mach numbers of 0.80 and 1.00 the tapered-in-thickness wing exhibits slightly higher drag due to lift than the constant 6-percent-thick wing.

3. The tapered 6- to 2-percent-thick wing exhibited an appreciably higher lift-curve slope above a Mach number of 0.85 and through the remaining test Mach number range than the constant 6-percent-thick wing.

4. Subsonic theoretical values of lift-curve slope, aerodynamic center, and lateral center of lift are in fairly good agreement with experiment. Agreement between theory and experiment is generally poor at the supersonic Mach numbers investigated.

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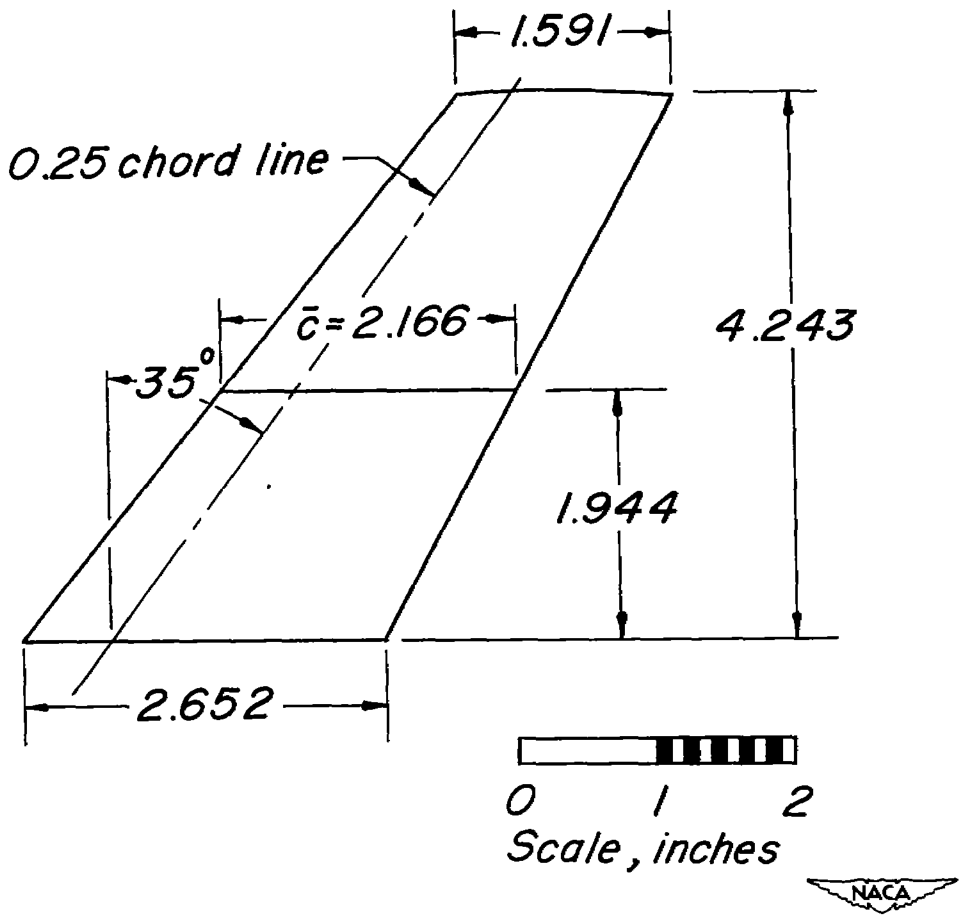
REFERENCES

1. Morrison, William D., Jr., and Fournier, Paul G.: Effects of Spanwise Thickness Variation on the Aerodynamic Characteristics of 35° and 45° Sweptback Wings of Aspect Ratio 6. Transonic-Bump Method. NACA RM L51D19, 1951.

2. Sleeman, William C., Jr., and Becht, Robert E.: Aerodynamic Characteristics of a Wing with Quarter-Chord Line Swept Back 35°, Aspect Ratio 4, Taper Ratio 0.6, and NACA 65A006 Airfoil Section. Transonic-Bump Method. NACA RM L9B25, 1949.

3. Donlan, Charles J., Myers, Boyd C., II, and Mattson, Axel T.: A Comparison of the Aerodynamic Characteristics at Transonic Speeds of Four Wing-Fuselage Configurations as Determined from Different Test Techniques. NACA RM L50H02, 1950.



Tabulated Wing Data

Area (Twice semispan)	0.125 sq ft
Aspect ratio	4
Taper ratio	0.6
Airfoil section parallel to free stream.	NACA 65A006 at root to NACA 65 A 002 at tip

Figure 1.- Plan-form drawing of a wing having 35° of sweepback, aspect ratio 4, taper ratio 0.60 and an NACA 65A006 airfoil section at root chord tapered to an NACA 65A002 airfoil section at tip chord.

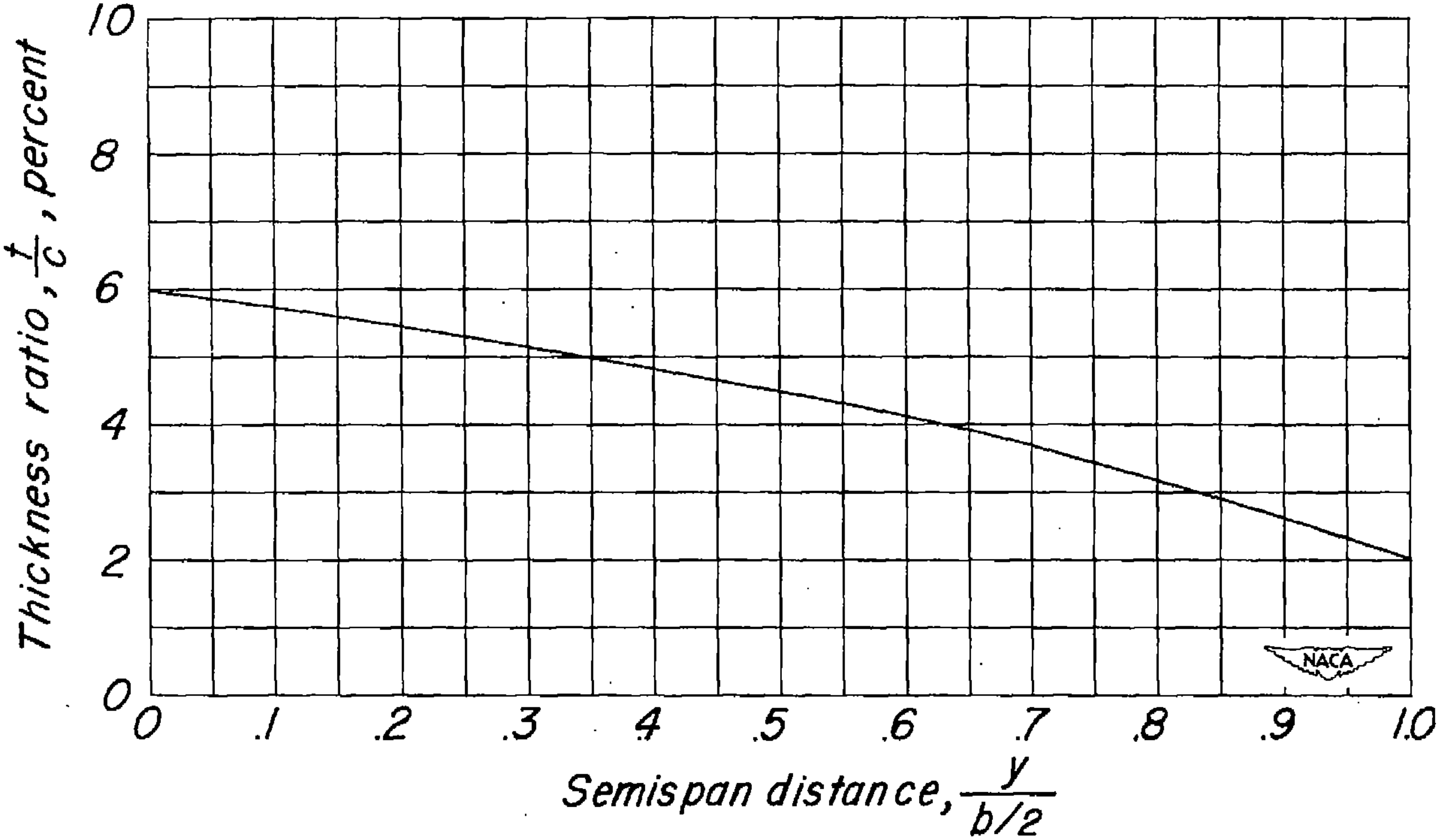


Figure 2.- Thickness-ratio distribution along semispan of a wing having 35° of sweepback, aspect ratio 4, taper ratio 0.60 and an NACA 65A006 airfoil section at root chord tapered to an NACA 65A002 airfoil section at tip chord.

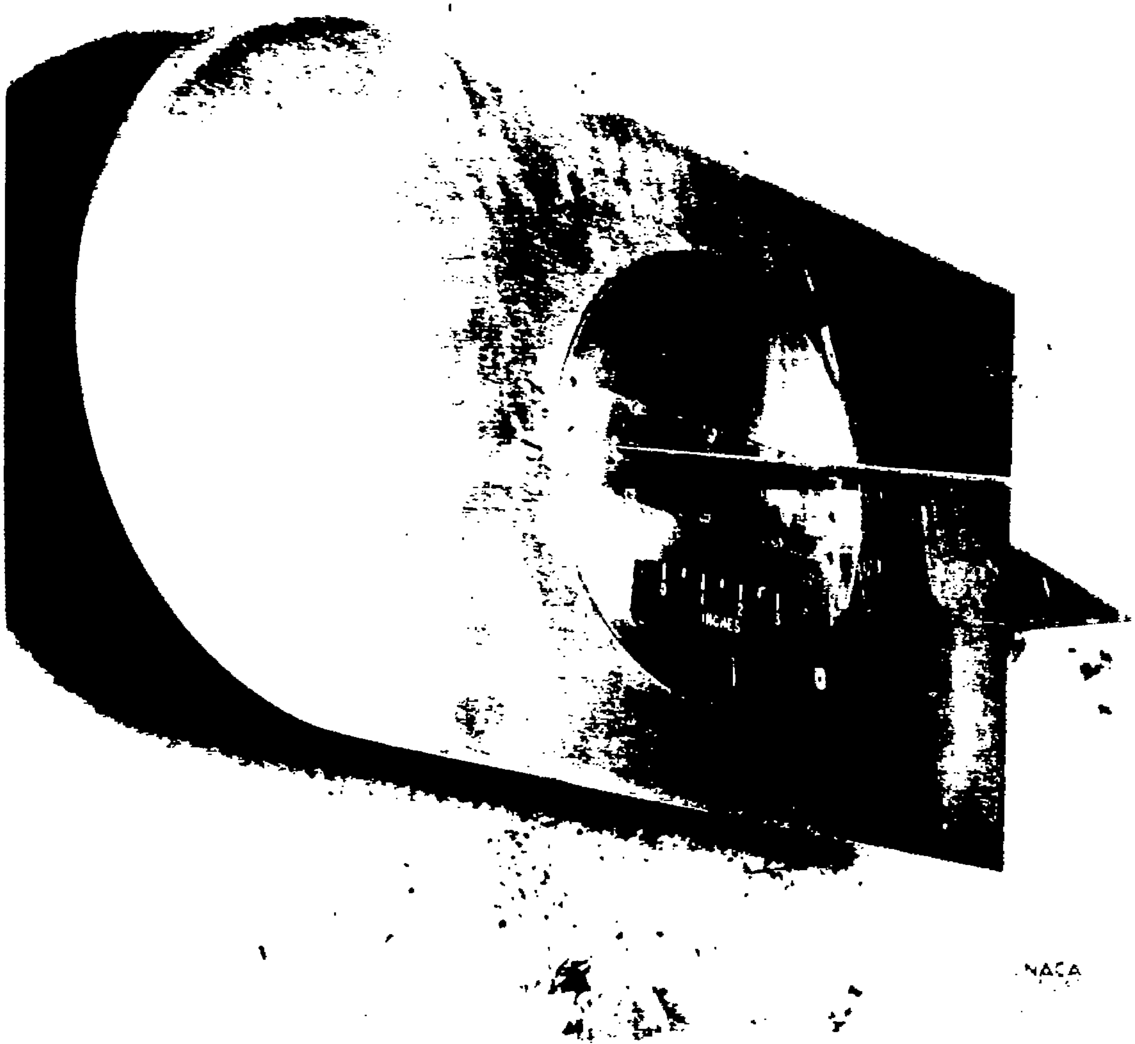


Figure 3.- Photograph of a wing on reflection-plane setup.

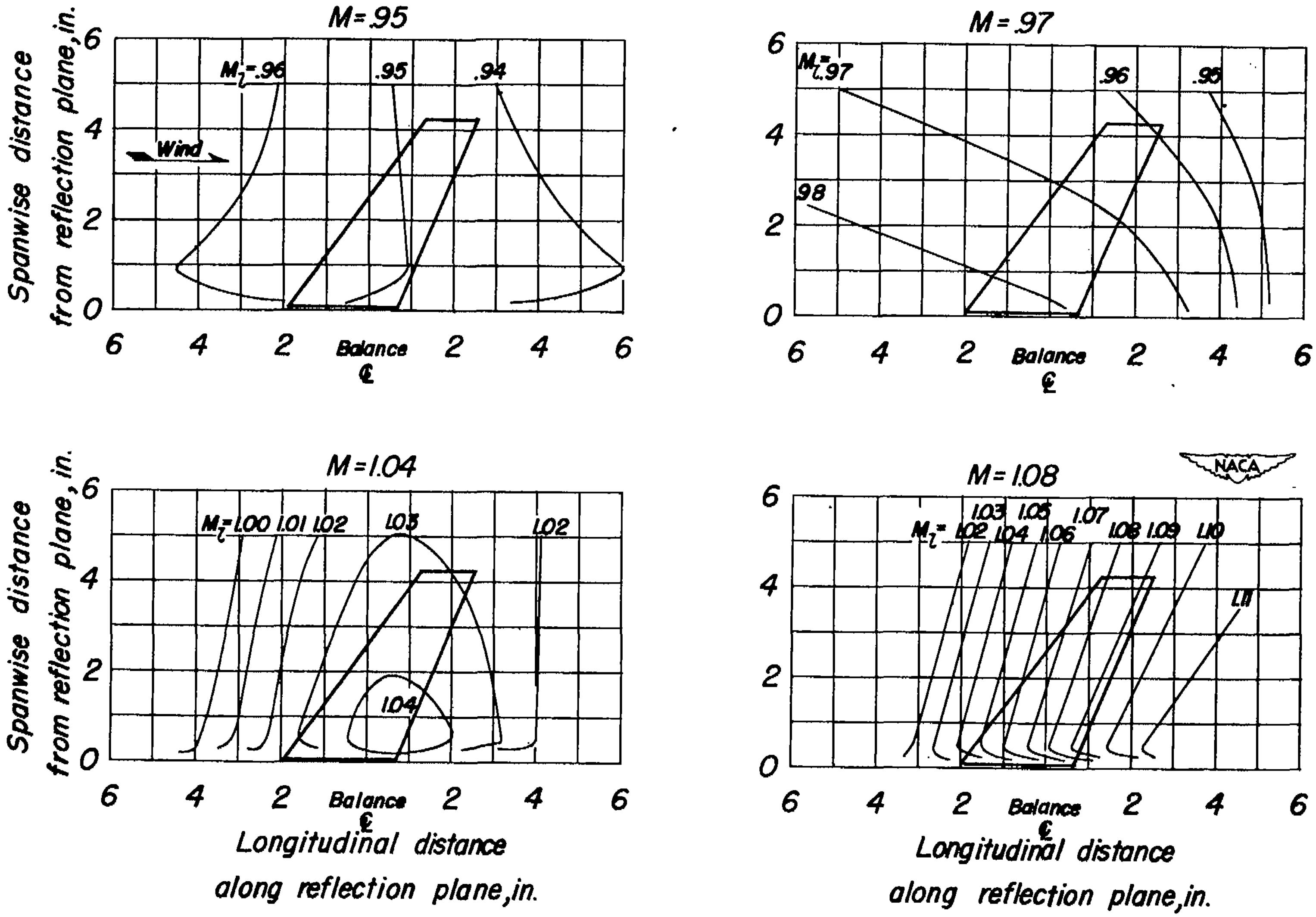


Figure 4.- Typical Mach number contours over side-wall reflection plane in region of wing location.

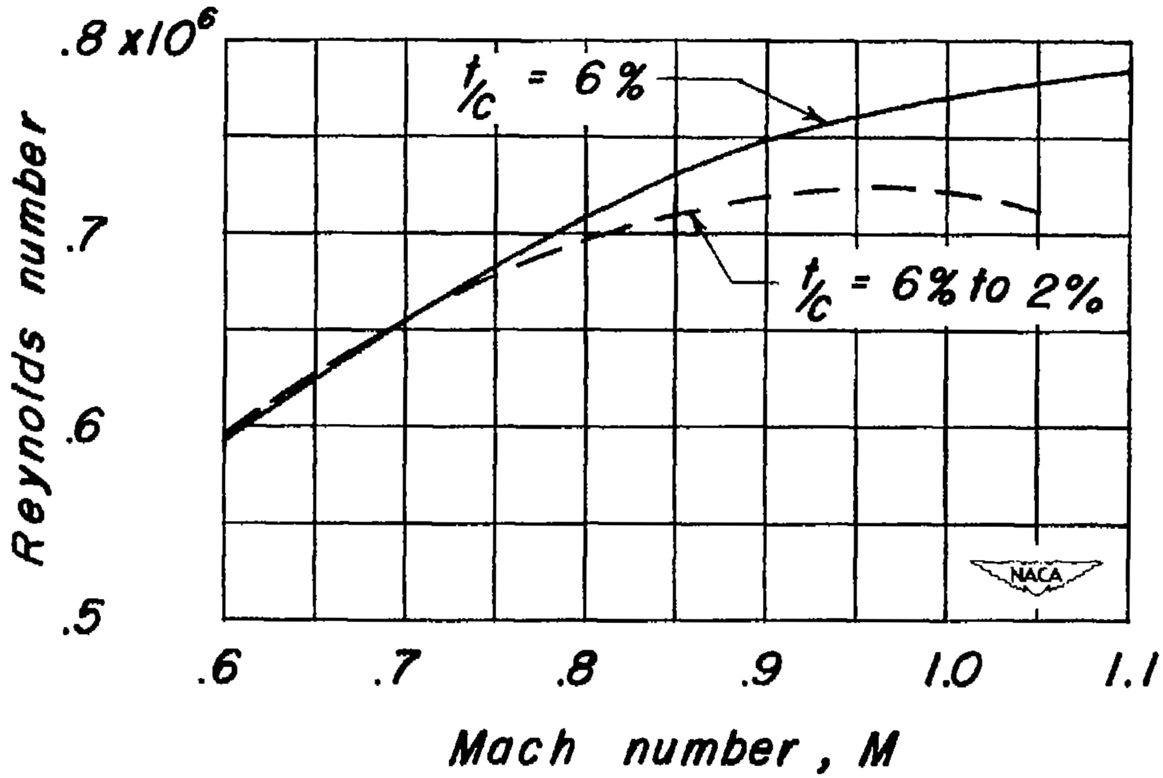


Figure 5.- Variation of average Reynolds number with Mach number for wings having 35° of sweepback, aspect ratio 4, and taper ratio 0.60.

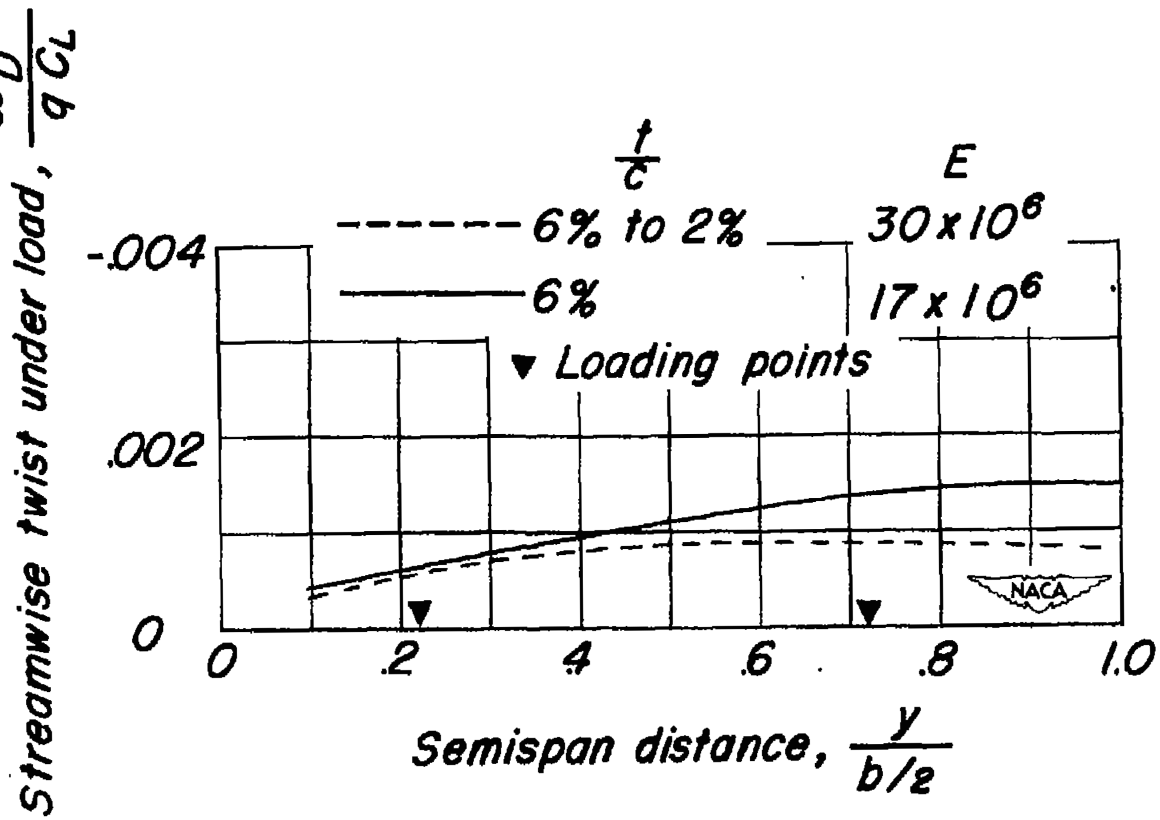
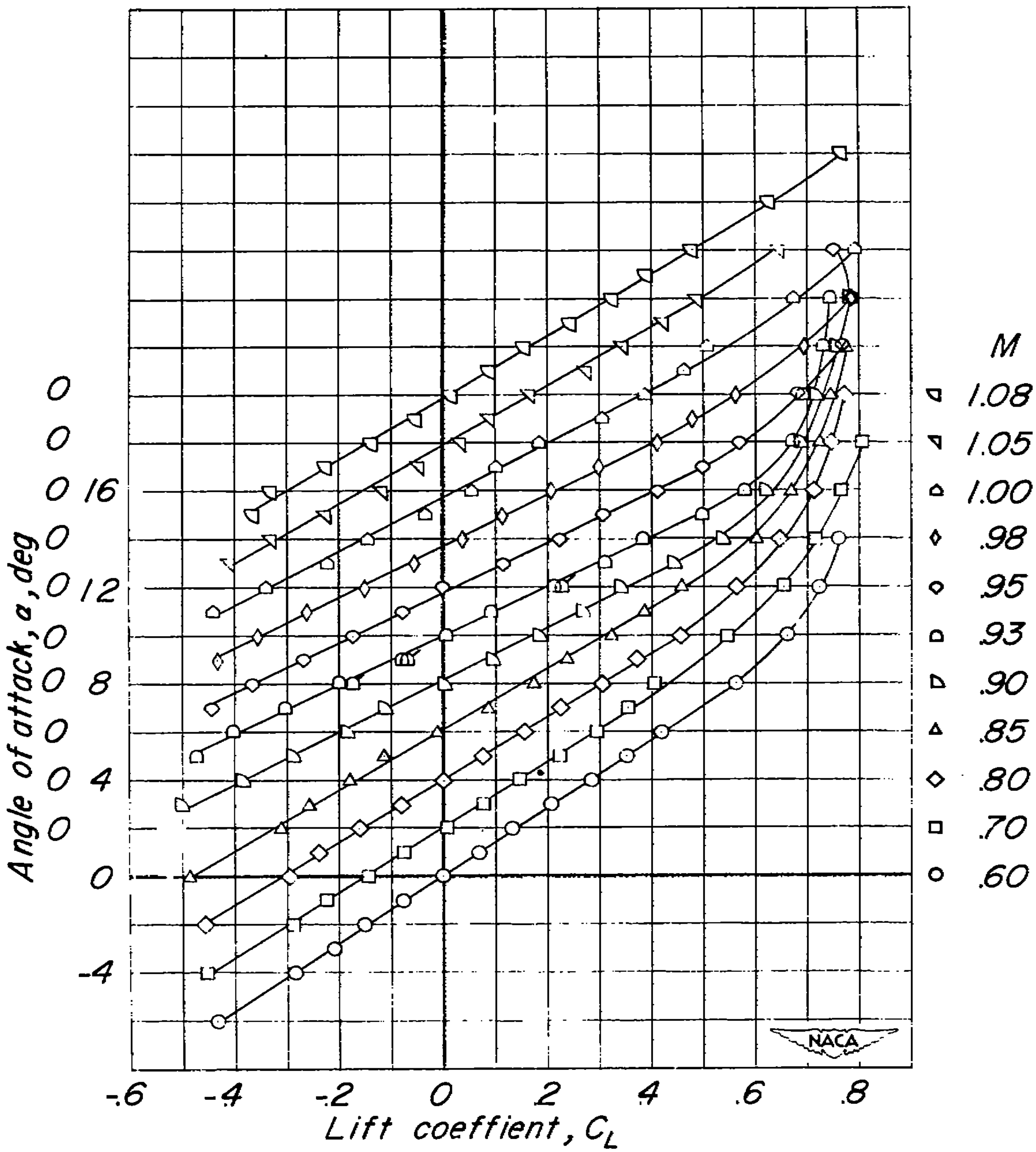
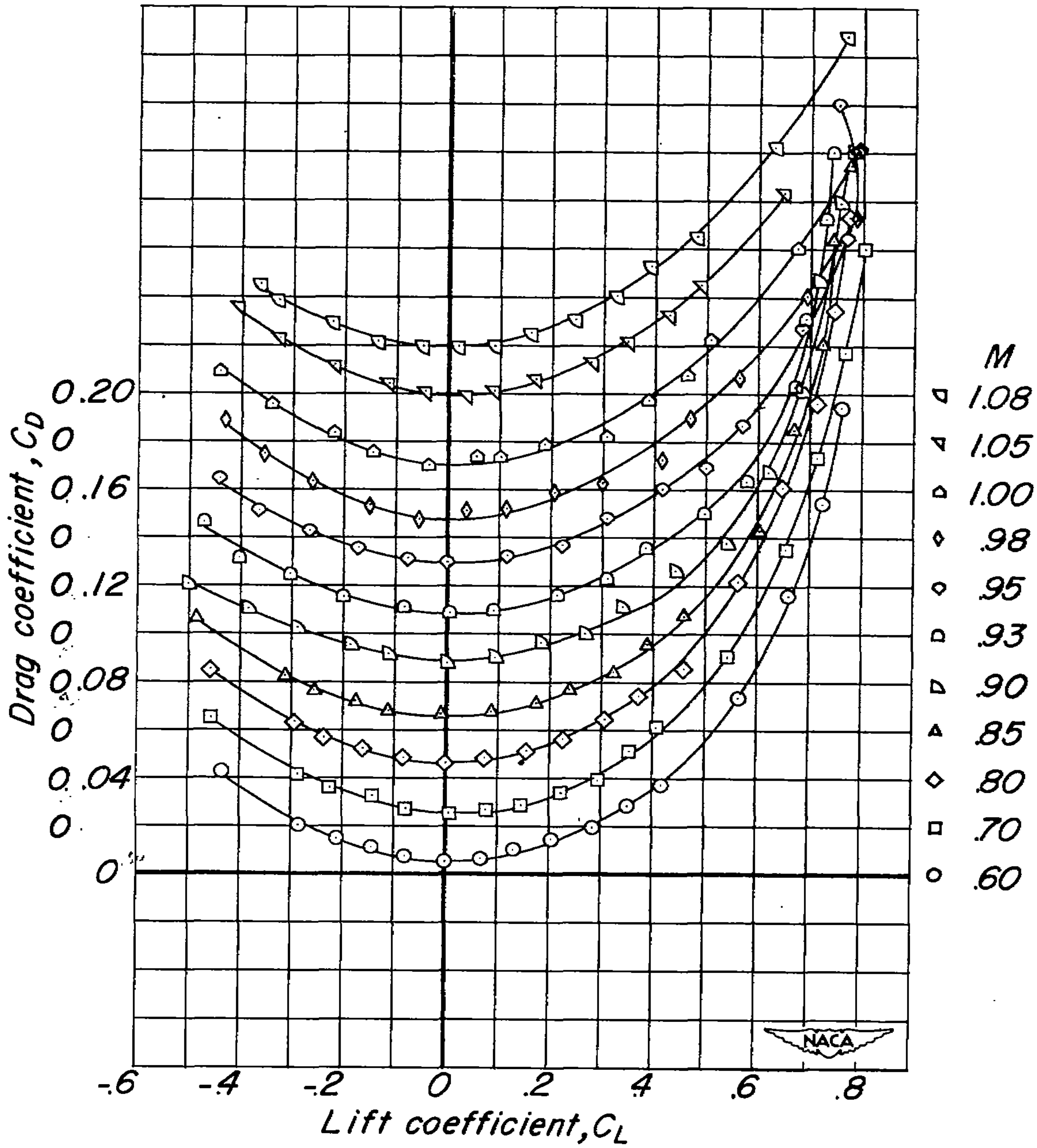


Figure 6.- Effect of thickness on variation of angle of streamwise twist along semispan for wings having 35° of sweepback, aspect ratio 4, and taper ratio 0.60.



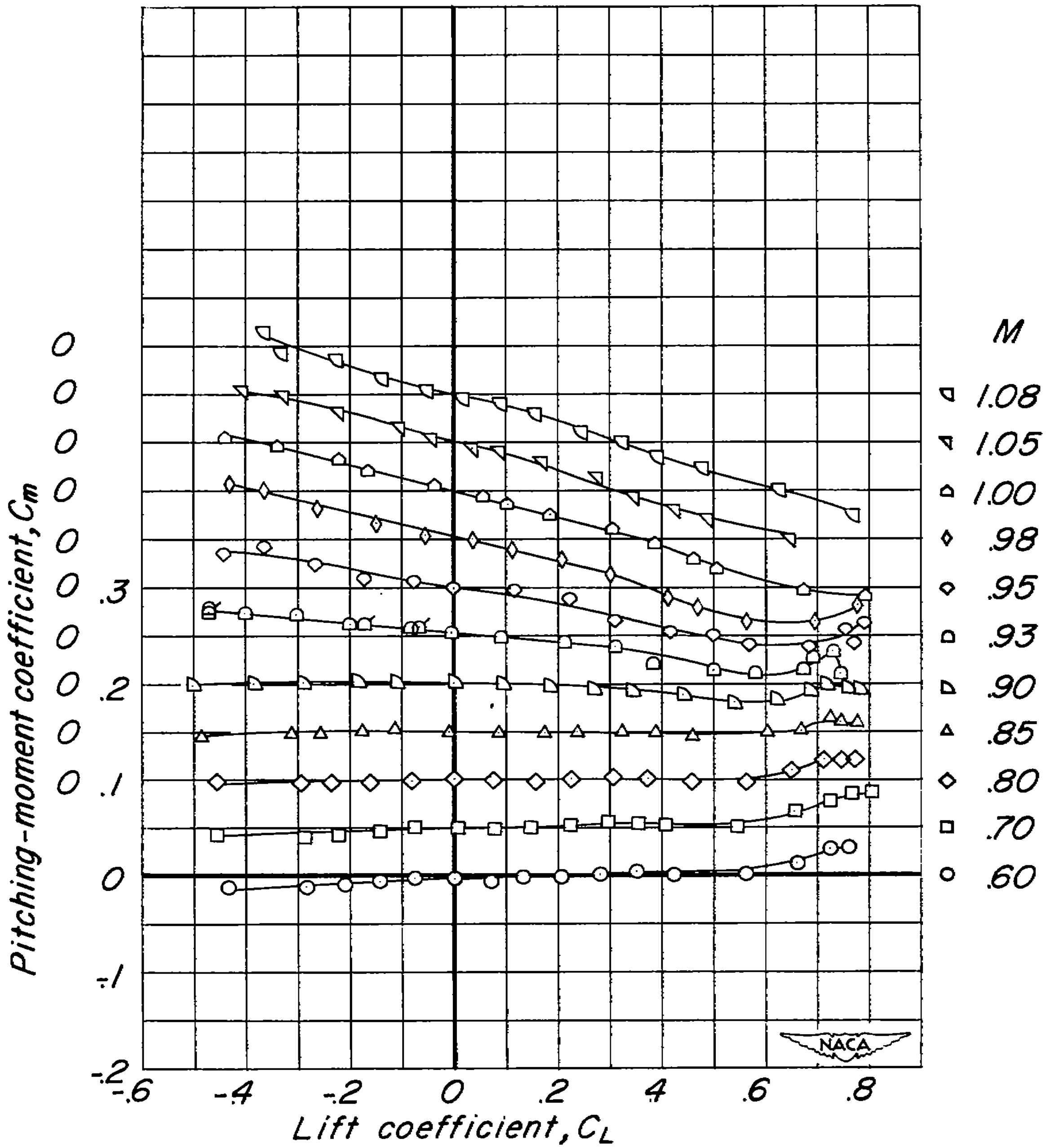
(a) α against C_L.

Figure 7.- Basic aerodynamic data for a wing having 35° of sweepback, aspect ratio 4, taper ratio 0.60, and NACA 65A006 airfoil section at root chord tapered to NACA 65A002 airfoil section at tip chord. Flagged symbols are check points.



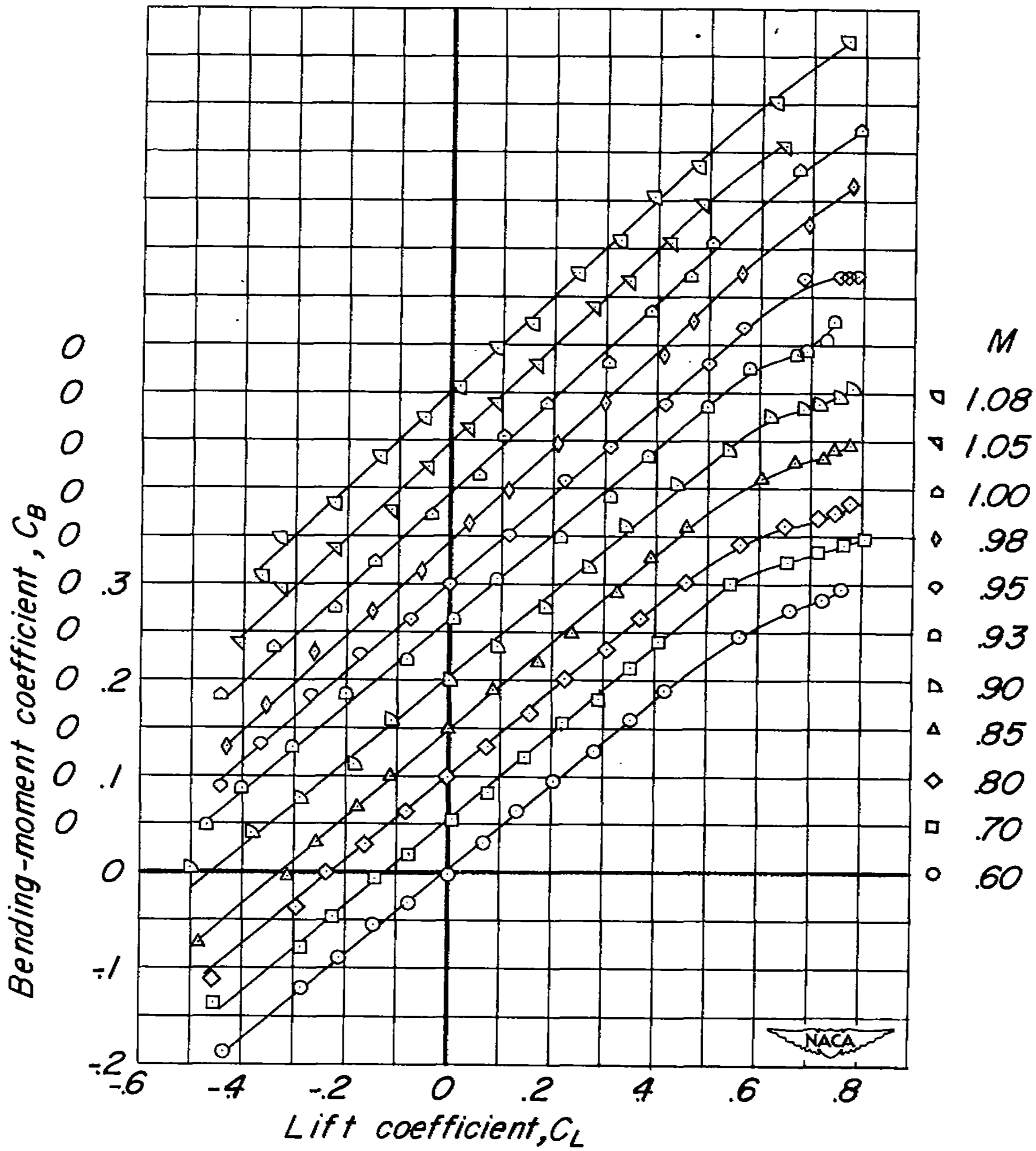
(b) C_D against C_L.

Figure 7.- Continued.



(c) C_m against C_L.

Figure 7.- Continued.



(d) C_B against C_L.

Figure 7.- Concluded.

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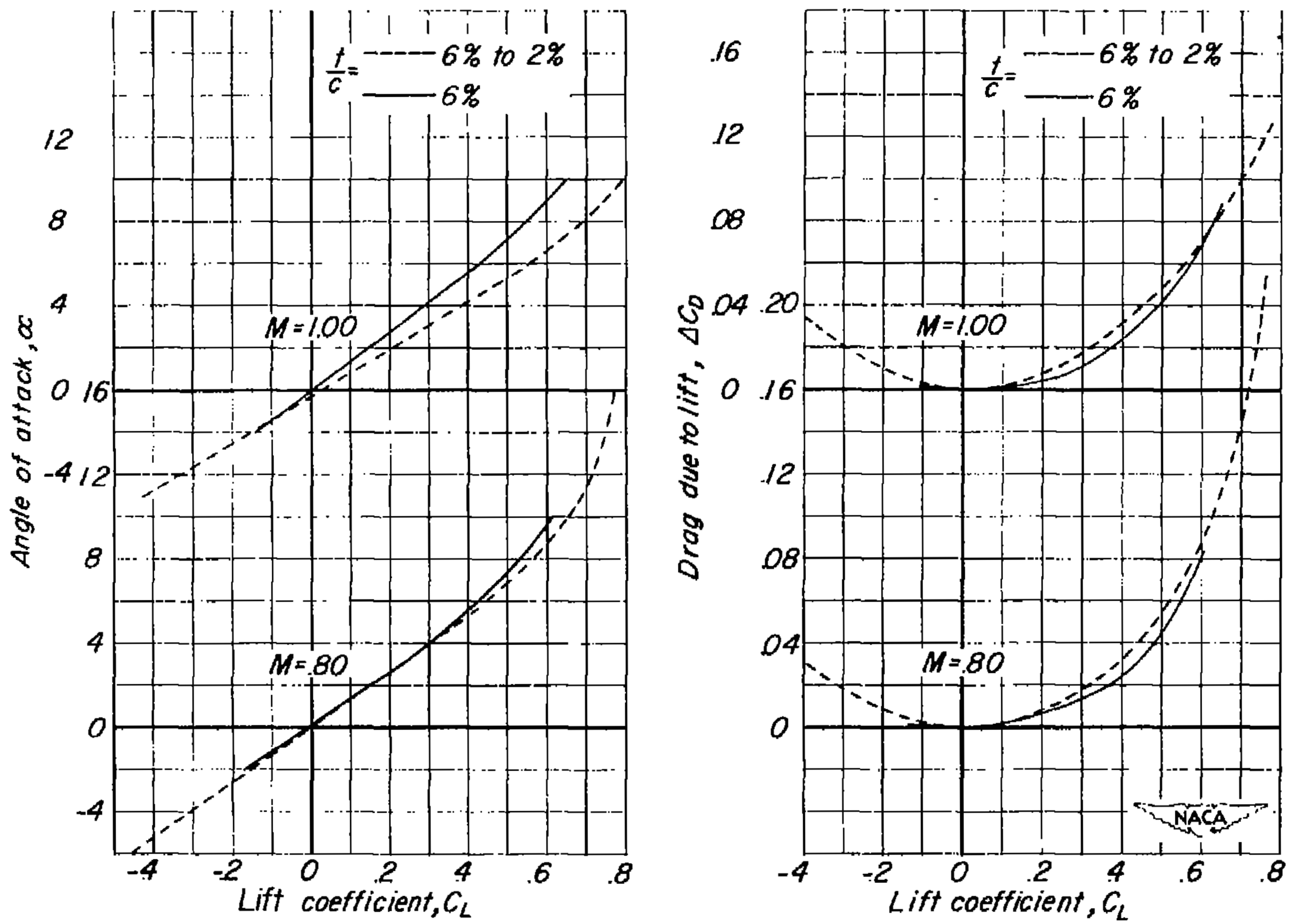
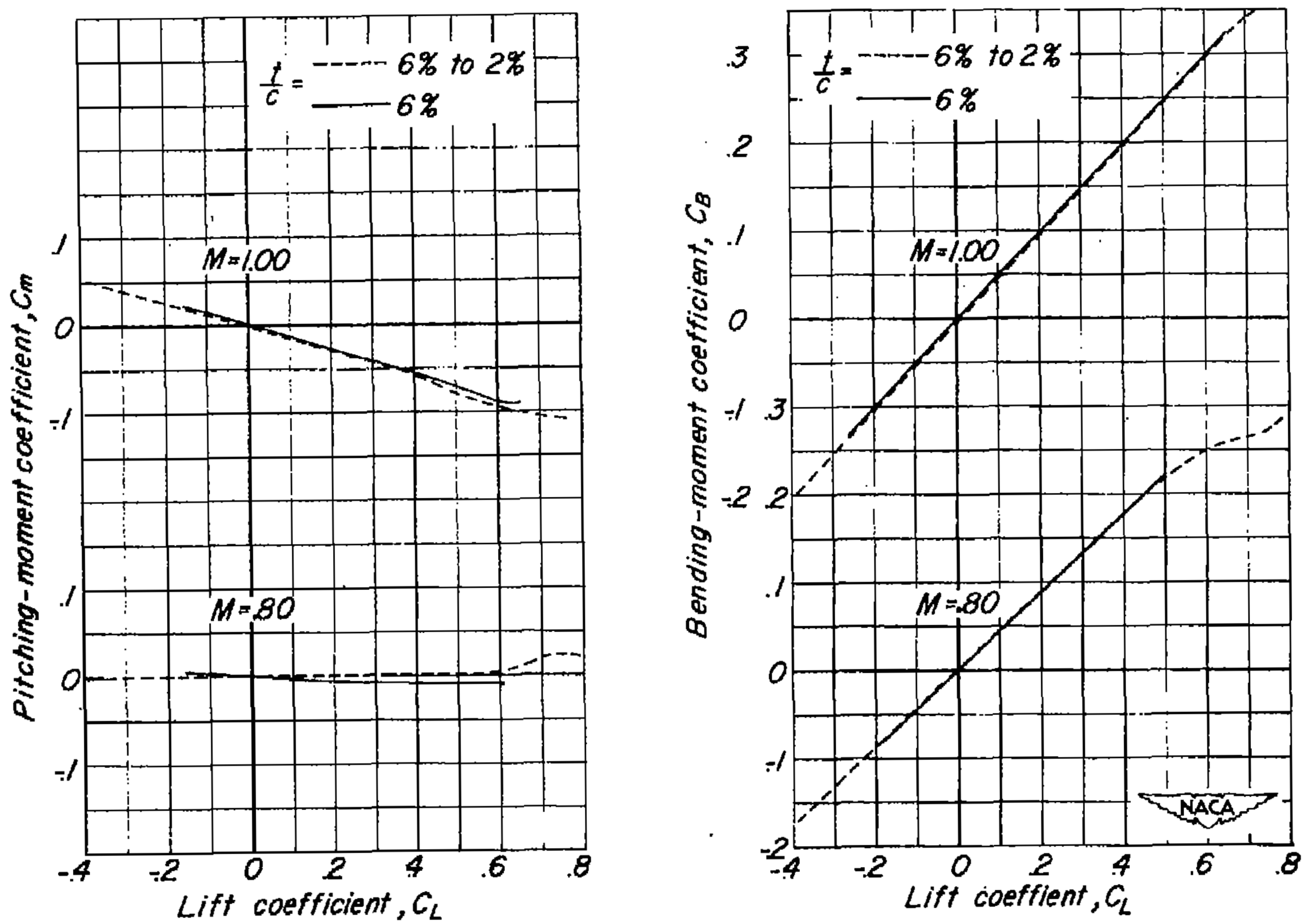


(a) α against C_L.

(b) ΔC_D against C_L.

Figure 8.- Comparisons at representative Mach numbers of the effects of thickness on the aerodynamic characteristics of wings having 35° of sweepback, aspect ratio 4, and taper ratio 0.60.



(c) C_m against C_L.

(d) C_B against C_L.

Figure 8.- Concluded.

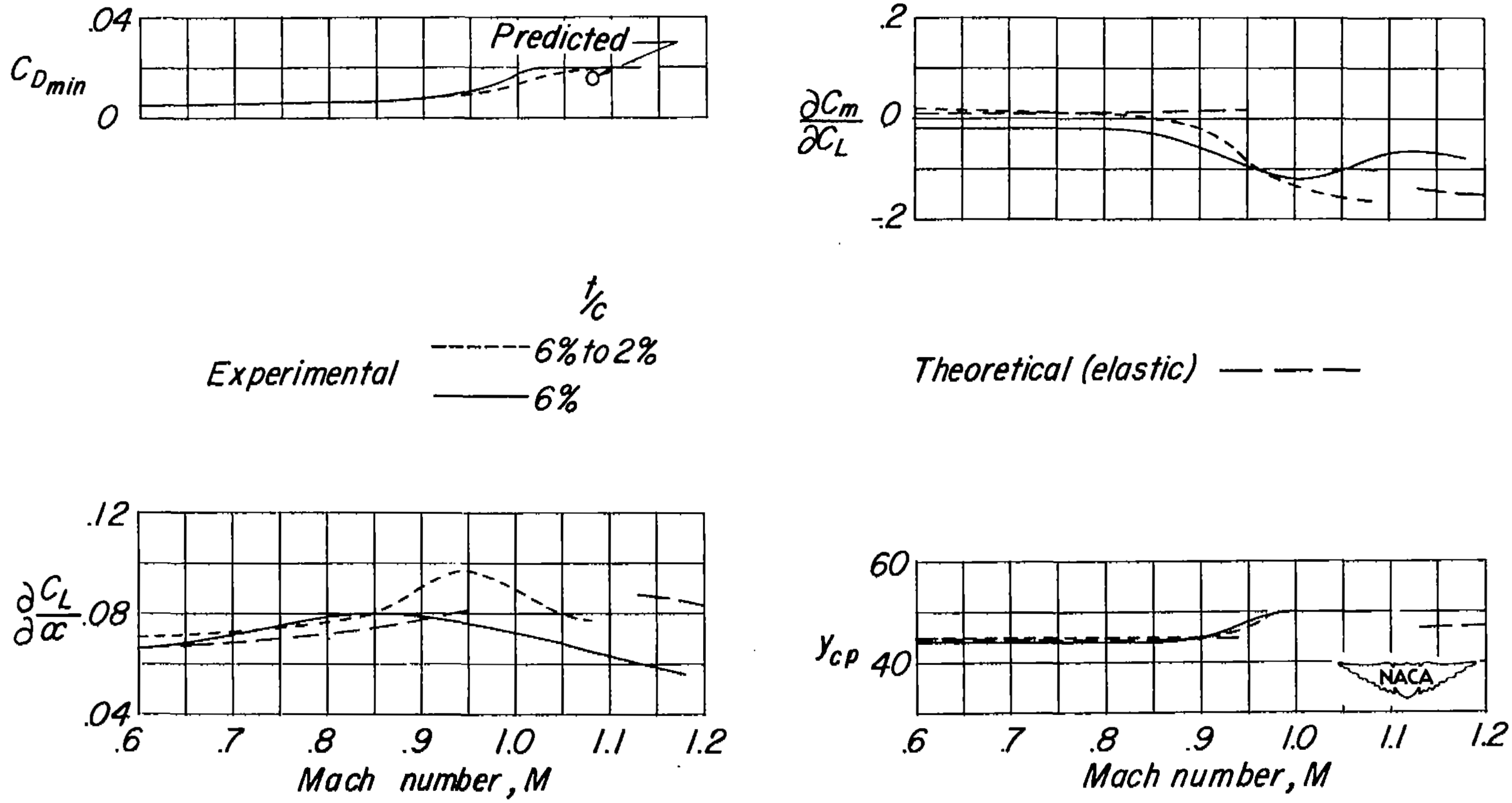


Figure 9.- Theoretical and experimental comparisons of the effects of thickness on the aerodynamic characteristics of wings having 35° of sweepback, aspect ratio 4, and taper ratio 0.60.

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